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NO. LED-510-4 #4

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PROGRESS REPORT -

TM-2 Thermal Analysis [U]

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I. INTRODUCTION

This is the first of a series of reports describing the thermal analysis of the TM-2 test vehicle and the thermal requirements for the TM-2 test program. Since this is the initial report some background information on the TM-2 program and the description of the test vehicle have been included in Sections II and III.

Sections IV and V present the thermal analysis for the external thermal sources and the internal thermal sources and sinks respectively.

The thermal requirements for the test program are discussed in Section VI. The detailed test plans will be published under separate cover and will be included in LPL-600-1, The Test Plan for the Lunar Excursion Module - Project Apollo. An initial TM-2 test plan will appear in the 15 August 1963 issue.

II. PURPOSE

The purpose of the TM-2 program is to provide an opportunity for evaluating the LEM thermal control system utilizing a full-scale model at an early date in the LEM development program. Specifically the test program will verify expected temperatures on the vehicle skins, pressure shell, primary structure, equipment, stored fluids and coolant loop.

In this way the actual performance of the vehicle insulation as well as values for contact resistance, radiation coupling and conduction coupling will be determined. The effectiveness of tankage and equipment mounting and insulation schemes as well as the effect of these equipments on the overall vehicle thermal balance will be determined. The test program will provide an opportunity to establish the portion of the actual equipment heat loads being carried away by the glycol cooling loop, those portions of the loads transmitted to adjacent equipment, and the effect of repositioning equipment on the cold plates on the coolant temperatures.

In addition, information will be provided on the cabin heat loads from external sources and from equipment mounted in the cabin and the variation in these values between pressurized and unpressurized cabin states. Specific information will also be obtained concerning the location of structural hot spots within the cabin due to the operation of equipment and the change in the temperature of these hot spots when the cabin is depressurized.

At present the design requirements for the LEM vehicle place few restrictions on the vehicle attitude (orientation) during the various mission phases (earth orbit, translunar, lunar orbit and lunar stay). Similarly the thermal control system must perform satisfactorily during a lunar stay which may occur at a point corresponding to any time of the lunar day or night. Thus, the TM-2 program must include testing over the combination of mission phases and vehicle attitudes which are thermally most severe for each of the subsystems.

To accomplish this evaluation of the thermal control system so that any necessary modifications may be made in the LTA-4 vehicle, an early test schedule has been established. At present the descent stage testing is to start in the Grumman vacuum chamber on 1 Feb. 1964 and the ascent stage testing is to start on 1 March 1964. To ensure that the engineering work is completed on time to have the vehicle ready for testing, a TM-2 vehicle team has been established. The responsibilities of the team members are presented in Reference 1.

III. TEST VEHICLE

To meet the objectives of the TM-2 program, the test vehicle must be a full-scale representation of the LEM which simulates the basic geometry, thermal properties and external environment thermal interface (Reference 2). The thermal paths, resistances, sources and sinks which will exist in the actual vehicle must be reproduced as accurately as possible.

In accordance with these requirements, the test vehicle will be a full-scale LEM consisting of ascent and descent stages based on the latest configuration drawings (LDW-280-10050 for the ascent and LDW-280-20000 for the descent). The structure will be prototype structure as it is known at the date of release of each drawing. The cabin will be capable of being pressurized to the design pressure of 5 psi with the outside pressure at 10^{-5} mm Hg and maintaining the specified leak rate of .2 #/hr.

Because of the early completion date for the vehicle, much of the production tooling will not be ready. Instead, the vehicle will be assembled with the best available tooling, and hand-welding will be employed for a large part of the pressure cabin.

All of the prototype insulation and the meteoroid shield will be installed as well as internal thermal control coatings. An external thermal control coating consistent with the vehicle-chamber requirements will be used. Experimental windows will be available for the pressure cabin.

To accommodate the vehicle in the thermal-vacuum chamber, the landing gear will not be included. A sketch of the TM-2 within the Grumman vacuum chamber is shown in Figure 1. A support system for the vehicle within the chamber is now being designed. The main propulsion and reaction control engines cannot be fired in the chamber. Consequently, thermal models of these engines will be employed rather than flight hardware.

All of the main propellant tanks will be included. However, they will be of a heavy weight aluminum design and not flight weight. The differences in their thermal response will be accounted for by modifying tank mounts. The tankage for all other fluids will be treated on an individual basis and may be installed or thermally simulated (See Section V). The problem of placing fluids in the tankage is complicated by considerations of personnel safety during the tests. The thermal responses of the prototype fluids will be duplicated by alternate fluids which are non-toxic and non-combustible.

For all tankage and equipment, the prototype mounting scheme will be employed including the use of actual insulating materials and coatings.

A live coolant loop for the electronic equipment will be used as well as experimental cold plates. In view of the early test dates, prototype hardware for the electronic equipment will not be available for the start of testing. Therefore, the TM-2 test program will commence with thermal simulators, designed and fabricated at Grumman, for all electronic equipment (See Section V).

The test wiring inside the vehicle will be laid out in a manner similar to that of the prototype wiring for power distribution. Where the bundles are not as large as prototype, dummy wiring will be added to provide the correct product of mass and specific heat.

IV. EXTERNAL THERMAL INPUTS

In order to properly test TM-2, the major external thermal inputs must be accounted for. A typical listing would include:

insolation
lunar albedo
lunar emission
plume effects
conduction from the surface of the moon
conduction and radiation from the C/M
earth emission and earth albedo

It is evident that not all of the aforementioned sources are concurrent; however various combinations must be readily available. Actual simulation including solar simulation, a lunar reflecting and emitting surface, an earth source, etc., is prohibitively expensive and complicated for this early stage; therefore TM-2 external inputs will be duplicated by the use of electrical heaters attached to the skin. This technique has been previously used at Grumman on the OAO vehicle.

It has been determined that the maximum power requirements are 7,900 watts for the ascent stage and 8,800 watts for the descent stage heaters. These values apply for a thermal control coating with an average solar absorptance of 0.1 and average infrared emittance of 0.5 with the LEM landing at the lunar subsolar point. In order to provide a realistic temperature distribution, these power levels are not distributed uniformly over the vehicle surface. Instead, the skin is divided into a number of areas of uniform flux absorption. The values of these fluxes are then determined analytically. The initial study was performed on a mathematical model consisting of forty-one surface elements. Calculations on this model produced the following information:

1. Insolation view factors (F_s)
2. Configuration factors to space (F_{sp})
3. Configuration factors to the moon (F_m)
4. Inter-area configuration factors (F_1)

These first four items provide data on each element for use in 5, 6 and 7 below.

5. Parametric plots of mean skin and internal temperatures as a function of net power into the LEM for various combinations of thermal couplings, thermal control coatings and lunar landing locations. This aids in the determination of gross thermal control parameters for a successful design

6. Three dimensional skin temperature distribution for the most likely cases culled from (5.) above. These present a realistic picture of skin temperatures during portions of the mission.

7. Power inputs to each skin element for proper simulation of different mission phases. This is required since each area is programmed for a specific input determined for the particular test being run.

As a typical example, the input for a sunlight lunar landing is calculated as:

$$I = \left[G_s \alpha_s \left(F_s + a_m F_m\right) + \sigma \epsilon \epsilon_m F_m T_m^4 \right] A/3.412$$

where:

I	= power to heater (watts)
G_s	= solar constant (BTU/hr-ft^2)
α_s	= solar absorptance of skin subdivision
ϵ	= infrared emittance of skin subdivision
F_s	= subdivision's view factor to insolation
a_m	= lunar albedo (to solar spectrum)
F_m	= configuration factor of the subdivision to the lunar surface
σ	= Stefan Boltzmann constant (BTU/hr-ft^2-$^\circ$R^4)
ϵ_m	= infrared emittance of the lunar surface
T_m	= temperature of the local lunar surface ($^\circ$R)
A	= subdivision area (sq. ft.)

This or a similar calculation must be performed for each element for most possible mission phases since the program as it is presently envisioned, will provide for testing all phases, including earth orbit, translunar coast, lunar orbit and lunar stay. Some sample results are presented in Table 1 with reference to Figure 2. The values are for:

$\alpha_s = 0.1$ $\quad\quad$ $\epsilon_m = 0.95$

$\epsilon = 0.5$ $\quad\quad$ $G_s = 443$ BTU/ft^2

where:

θ = angle to the LEM-sun line measured from the +X axis.

γ = angle to the LEM-sun line, projected on the Y-Z plane, measured from the +Z axis.

F_1 = inter-area configuration factor

T = skin temperature of the particular element ($^{\circ}$R)

				$\theta = 0^{\circ}$				$\theta = 90^{\circ}$ $= 180^{\circ}$			
Area	F_{sp}	F_m	F_1	F_s	T	I	I/ft^2	F_s	T	I	I/ft^2
3	0.5	0.5	0	0	520	249	25.6	.05	343	38	3.9
4	1.0	0	0	1	475	468	13	0	165	0	0
28	0.5	0.5	0	0	520	511	25.7	1	532	336	16.9
33	0.8	0	0.2	1	512	90	13	0	316	0	0

TABLE 1

For the aforementioned coating (α_s = .1), heater power requirements vary from 3 to 30 watts per square foot depending on the specific skin area under discussion. The skin heaters and controls must be capable of providing this range. An extended input limit results from an increase in α_s, for values up to unity. For this extreme, the high power level would be approximately 200 watts per square foot. The probability of using a high absorptance finish is very small, however this testing capability has been included to cover any potential problem.

Recent changes in the configuration coupled with the need for a more accurate analysis have limited the applicability of the 41 element study to the present LEM.

In order to provide useful information, the new model consists of approximately two hundred surface elements. A more refined study is underway and it will furnish information similar to that already discussed in addition to internal temperature distributions. Part of the new data will be obtained from the 1/6 scale model test program mentioned later in this section. The details of this study will be reported in the near future.

Another facet of the preliminary work on TM-2 is the program to evaluate various types of skin heaters based on the following criteria:

1. Availability
2. Ease of application and removal
3. Uniformity of power distribution
4. Minimum temperature gradients and minimum thermal mass
5. Ability to be coated
6. Ability of mounting to withstand wide temperature variations.

The first two requirements are a result of the desire for an "in-house" capability to minimize down-time for new heater preparation. Such a delay might be due to a configuration change, a damaged heater or perhaps an engineering modification which obviates the previous heater. The next two characteristics are justified from a thermal standpoint in that the actual input to the skin would be uniform, create no temperature gradient outside the skin, or alter the thermal characteristics of the skin. In order to control the radiation from the skins to the chamber LN_2 shroud, the heater must be selectively coated on its outer face, hence the fifth criterion. Lastly, the heater must remain in good thermal contact with the skin during all phases of the tests.

As of the date of this writing a goodly number of possible heaters have been considered. They fall into the following major categories:

1. Preformed flexible blankets
2. Spray-on heaters
3. Resistive paints
4. Graphite sandwich materials
5. Woven cloth and wire mesh
6. Carbon cloth
7. Nichrome mesh
8. Nichrome ribbon

Except for 7 and 8, all of the categories have been eliminated because they failed to meet one or more of the evaluating criteria. For example, items one and two were eliminated because they could not be fabricated quickly, in-house; item three because of application difficulties. The remaining heaters are being tested, with various bonding agents, under extreme operating conditions to determine the best possible combinations. The result, so far, favors a heater consisting of a one mil nichrome ribbon bonded to a three and a half mil fiberglas cloth. This heater is sufficiently flexible to make application possible and elements can be cut to size from larger sheets which are kept in stock. The temperature gradient from the heater element to the skin is less than two degrees Farenheit during a typical transient test and the power density as determined by temperature distribution is satisfactory. Thus, at this early date, it appears that a successful heater is readily obtainable. Efforts towards this goal are being continued.

It should be noted that the skin heaters are not sufficient to complete the external simulation. Test chamber restrictions are such that vehicle appendages, e.g., reaction control quads and landing legs, will not fit into the working space available. However, mounting brackets for such equipment will be included in TM-2, and heaters, simulating the input to these brackets, will duplicate the thermal loads from the eliminated articles to the main LEM structure. These heater loads will be determined by an analysis considering all pertinent sources.

The one remaining requirement is the simulation of the thermal effects of space itself. In the Grumman vacuum chamber, liquid nitrogen shrouds are used to provide a radiation sink similar to space. For most mission phases, the 160°R temperature of the LN_2 provides a very satisfactory background, i.e., for vehicle skin temperatures of 350°R the error due to the simulation is only about 3°R. For lower skin temperatures, however, the error increases. If a lunar landing condition with no insolation is to be tested ($90 < \theta < 180$) skin temperatures down to 160°R are possible. At this level the maximum error approaches 50°R. An error of approximately 15°R is anticipated at 250°R. If the analysis shows that the optimum configuration and thermal control surfaces combine to give skin temperatures much below 250°R, testing with the currently available LN_2 chamber walls will provide only limited results.

Corrections to the subdivision power levels will be made based on the model/chamber geometrical relationships as well as temperature and thermal coating of the LN_2 shroud and the TM-2 (Reference 3). The error due to the 160°R background temperature has been discussed above, the two remaining major problem areas are the unwanted reflection of vehicle emission and the uncertainty in measured emittance values of the TM-2 skin and the cold wall. As a typical example, surface 28 (Figure 2) would have an error of 7 watts/square foot if its temperature during test was 700°R and its infrared emittance was 0.8. To achieve this temperature the basic power level would be approximately 35 watts/square foot. Note (in Figure 1) that this surface is very close to the chamber wall; therefore its associated error is large. The calculated input correction does not and indeed cannot compensate for uncertainties in the TM-2 emittance value, however a temperature tolerance will be applied to the test data.

In order to determine the power levels absorbed by a given surface subdivision, the view and configuration factors of the LEM must be determined. This represents still another pre-test program currently being conducted. The insolation view factor, which is straightforward for simple geometries, becomes complex when vehicle surface reflections are included in the analysis. In order to help solve this and other related problems, Grumman will make use of an instrumented scale model of the LEM (reference 4). For determining the view factors of each of the surface subdivisions a measure of the variation of incident light with vehicle orientation is required. The sensor for this test is an uncoated, ungridded solar cell chosen because the short circuit output is proportional to the cosine of the angle of incidence of the illumination as well as its intensity (Reference 5). By obtaining the ratio of cell output at some arbitrary orientation to that of normal light incidence, the view factor is determined directly. In addition, this determination includes the effect of reflections from adjacent surfaces since the model is painted to duplicate the actual thermal control surface. In practice each cell represents one subdivision, and by testing over a complete range of vehicle orientations, a tabulation of view factors for each surface subdivision is developed. Since relative geometry (and not size) is the criteria, a scale model of the LEM is used. This mockup is 1/6 scale and made of wood, segmented for ease in modification. The collimated light source is a reworked, five foot diameter, carbon arc searchlight.

This same model, when instrumented with calorimetric sensors, will provide the configuration factors to space. To do this the model will be mounted in a vacuum chamber with a cold wall simulating space. When illuminated with a solar source, the temperatures of the sensors will be indicative of their respective space configuration factors. By testing at various orientations to the light source, a tabulation of these factors will be generated. The configuration factor to the lunar surface is the remaining factor required to complete calculations of the power inputs. These lunar values are calculated using standard configuration factor evaluation techniques.

V. INTERNAL THERMAL LOADS

To provide a satisfactory test vehicle, all of the internal heat sources and skins must be included. The glycol cooling loop will be a live loop with experimental cold plates. For the prototype LEM, the glycol loop will reject its heat to a water boiler. However, for TM-2, the glycol lines will be brought outside the vacuum chamber to a cart whose cooling and pumping capabilities simulate those of the prototype ECS (Environmental Control Subsystem) glycol pumps and water boiler.

The life support loop of the ECS will not be included. Those portions of the metabolic heat loads which are vented to the cabin and picked up by the cabin heat exchanger will be simulated by heaters. The cabin fan and cabin heat exchanger will be included as operating equipment.

Since actual LEM electronic and electrical power supply equipment will not be available in time for the start of the TM-2 test program, thermal models or mockups of the equipment will be used. These models will be the least complex simulation necessary to obtain useful test results. As a minimum, the models will simulate the actual equipment's thermal dissipation, geometry, structural and/or cold plate mounts, mass-specific heat product, and infrared surface properties. However, no attempt will be made to reproduce the equipment internal temperature distributions and structure (i.e. thermal diffusivity) since this type of simulation would be extremely complex and is not necessary for an overall vehicle thermal test.

The thermal models will be designed and fabricated at Grumman. The equipment form factor, mass, and mounting procedure will be based on the current LEM equipment arrangement and packaging concepts. External equipment surfaces will be coated with appropriate paints to obtain the desired infrared emittance. Non-electronic items, such as tankage containing cryogenic fluids (SOX, GOX, SH_2), cabin lights, major propulsion valves, cabin displays and controls, and cryogenic heat exchangers are significant and will be thermally simulated. Simulation of the Inertial Measuring Unit (IMU) will be obtained by the addition of heaters and thermal mass to an IMU case being obtained from MIT.

The simulation of the equipment thermal dissipations will generally be obtained by the use of two 115 volt AC electrical heaters. These heaters will be sized such that either heater will be capable of proper simulation in the event of failure of the other. In cases where local hot spots in the packages result in uneven thermal inputs to the cold plates, more than two heaters will be required to achieve the desired simulation. (e.g. the MIT computer and Power Servo Assembly which must be simulated at the tray and stick levels respectively. The thermal dissipation of any low powered pieces of equipment will be neglected. The thermal dissipation from the fuel cells and the hydrogen vent line from the fuel cells will be simulated by temperature control of the external surfaces of these items. Wherever possible, the equipment electrical duty cycles will be approximated by steady state heat inputs. When this method is unacceptable, quasi-steady state or transient heater inputs will be used.

Table 2 indicates the principle units which will be simulated in the TM-2 vehicle.

TABLE 2

SIMULATED EQUIPMENT FOR TM-2

Instrumentation Subsystem

Pulse Storage Equipment
Pulse Code Modulator
Digital Timing Unit
In-Flight Test System
Signal Conditioners
Display Transducers
IFTS Test Transducers
IFTS Caution Transducers
Scientific Instrumentation (GFE)

Navigation and Guidance Subsystem

Radar Altimeter
Transponder
Rendezvous Radar
Inertial Measuring Unit (GFE)
Optical Measuring Unit (GFE)
Apollo Guidance Computer (GFE)
Power Servo Assembly (GFE)
Coupling and Display Unit (GFE)

Reaction Control

Thruster Solenoid
Ascent Interconnect

Stabilization and Control Subsystem

Rate Gyro Assembly
Attitude Reference Assembly
Attitude and Translation Control Assembly
Descent Engine Control Assembly
Guidance Coupler Assembly
Computer
Attitude Controller
Thrust Controller
Panel
Backup Guidance Platform

Communications Subsystem

Audio Center
S-Band Transceiver
VHF Transceiver
VHF Diplexer
Premodulator Processor
S-Band Power Amplifier
S-Band Diplexers
S-Band Switch
VHF Switch
S-Band Steerable Antenna and Drive and Electronics

Electrical Power Subsystem

Fuel Cells
Fuel Cell - ESS
H_2 Vent Lines
Inverters
Converters
Battery Charger
Spiking Battery
Emergency Battery
Peaking Battery
Relay Boxes
Parasitic Loads
Distribution Lines
GOX Tanks
SOX Tanks
SH_2 Tanks

Crew Systems

Cabin Lights
Display Panel Lights
Docking Lights
Instrumentation Lights
Warning Lights
Digital Readout
Thumb Wheels

Propulsion - Descent

Actuator - Gimbal
Amplifier
Propellant Solenoid Shut-off Valve
Helium Injection Valve

Displays and Controls

Propulsion

Warning Lights
Servos

Flight Controls Panels (2)

Mode Light
Servos
Crossed Pointer Readout
Flight Director Attitude Indicator
EL Lighting

ECS Panel

Warning Lights
Servos

N & G Panel

MIT(GFE)
Back-up Guidance

RCS

Servos
Warning Lights

Displays and Controls (Cont)

Thrust Control Panel

Warning Lights
Digital Readout

Clock

IFTS

Lights

Caution Panel

Lights

Due to the lack of information from NASA concerning the GFE Scientific Equipment it may not be possible to simulate this equipment in the descent stage.

Detailed information concerning the heater sizing for these models will be presented at a later date.

In addition to the vehicle primary structure, there are other heat sinks which must be included in TM-2. Propellant tanks for the ascent and descent engines will be of the heavyweight design and will be filled with referee fluids. The reaction control propellant tanks will also be loaded with referee fluids. The helium tanks for pressurization will be included but will not be loaded with a gas. Lines and thermally significant valves will be installed or modeled but no attempt will be made to run fluids through them.

The water tank will be filled with a thermally satisfactory fluid which will not present a freezing problem should the test temperature fall below the expected minimum.

The ascent and descent engines will be thermally modeled and the engine heat fluxes into the stage structure will be produced with electrically powered heaters similar to the skin heaters for external thermal inputs.

VI. TEST CONDITIONS

It has already been noted that the capability for testing any mission phase will exist. However, preliminary analysis indicates that the widest vehicle temperature extremes will be experienced during lunar stay at various possible landing locations and during translunar coast. Therefore, initial tests will be steady state simulations of the aforementioned portions of the flight. These will cover ranges of θ and γ sufficient to present an adequate worst case temperature profile. A typical set of runs would start with conditions at $\theta = 180^\circ$ and run through $\theta = 90^\circ$ to $\theta = 0^\circ$ with runs at varying γ values interspersed wherever appropriate. In addition, the translunar orientations which result in worst case conditions (e.g. cold soak of descent engine, RCS, etc.) will be duplicated. In order to use the chamber time efficiently, tests will probably start at the coldest level, since heating the vehicle is faster than cooling it. However, it is intended that TM-2 testing shall remain flexible. Any particular condition which seems to present difficulties will be closely investigated. If gross inadequacies in the thermal control system are revealed they will be corrected before testing continues. Upon completion of the first series, attention will be directed towards cyclical mission phases. For these phases, e.g., lunar and earth orbit, the heater inputs will be determined by calculating the mean power levels over the cycle under consideration. Those values which pertain to worst case conditions will, in turn, provide extreme mean temperature limits for the LEM. Since the thermal time constant of the vehicle is much larger than the orbital periods, mean temperatures are acceptable. As the third and final series, transient tests will be run if deemed advisable. Inputs will be programmed in real time to provide information about short phases such as lunar descent and ascent, special purpose translunar maneuvers, etc. No attempt will be made to duplicate the extremely high temperatures expected on engine nozzles and related equipment. Instead, heaters will be used on adjacent shielding which is at a reasonable temperature, as demonstrated by analysis. It is expected that series one will take up the bulk of the test time since it does provide data most pertinent to the overall thermal control system.

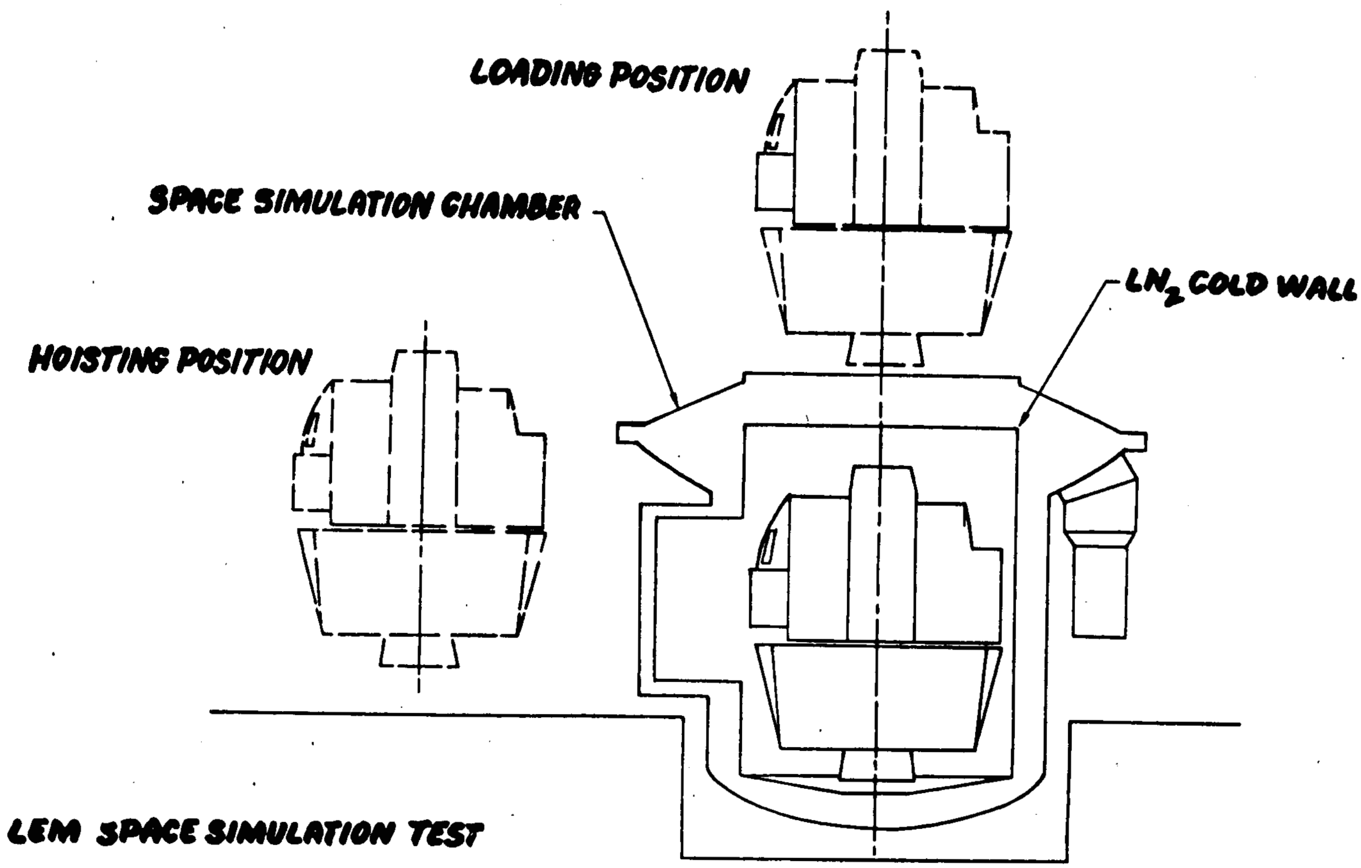
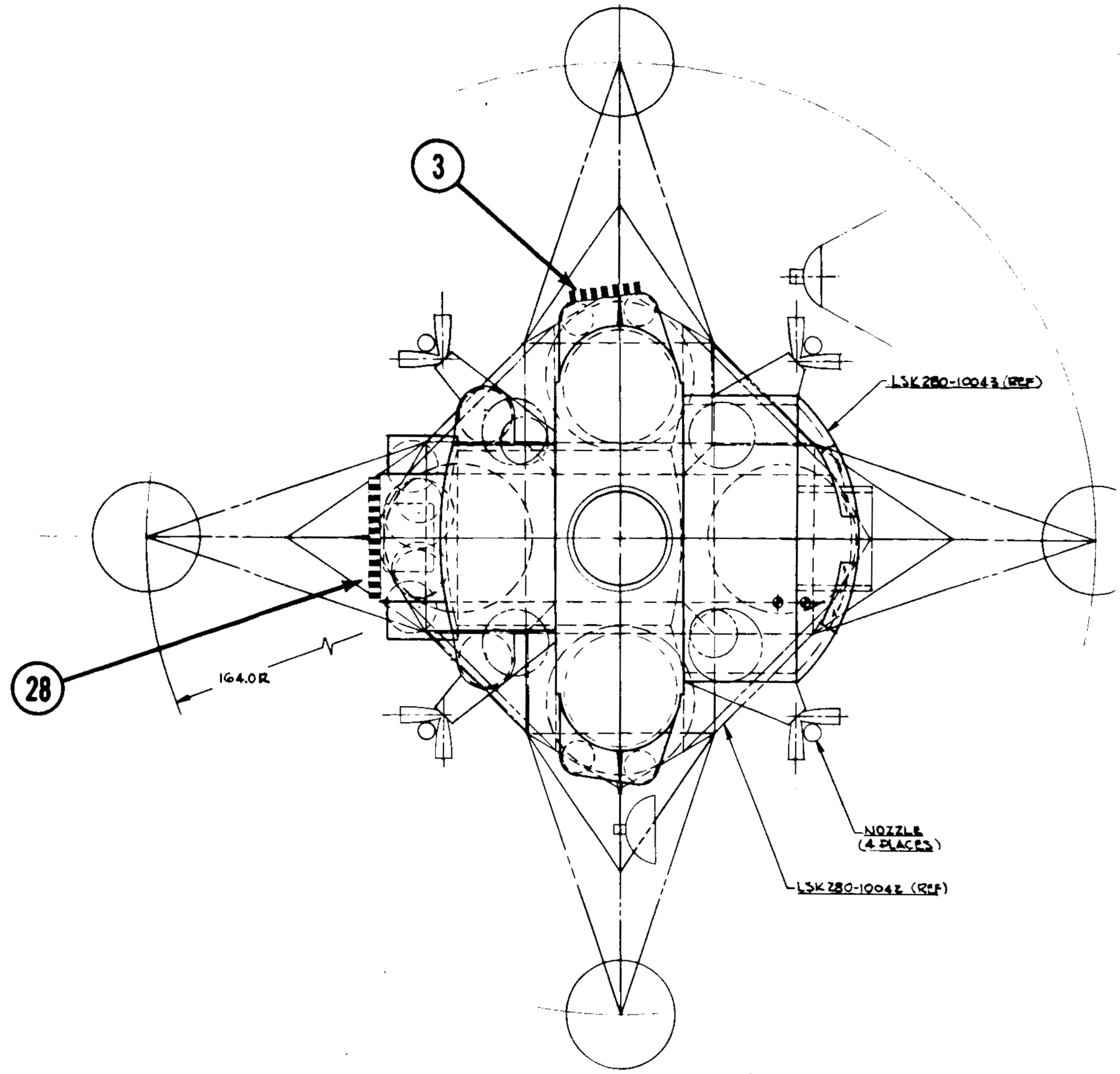
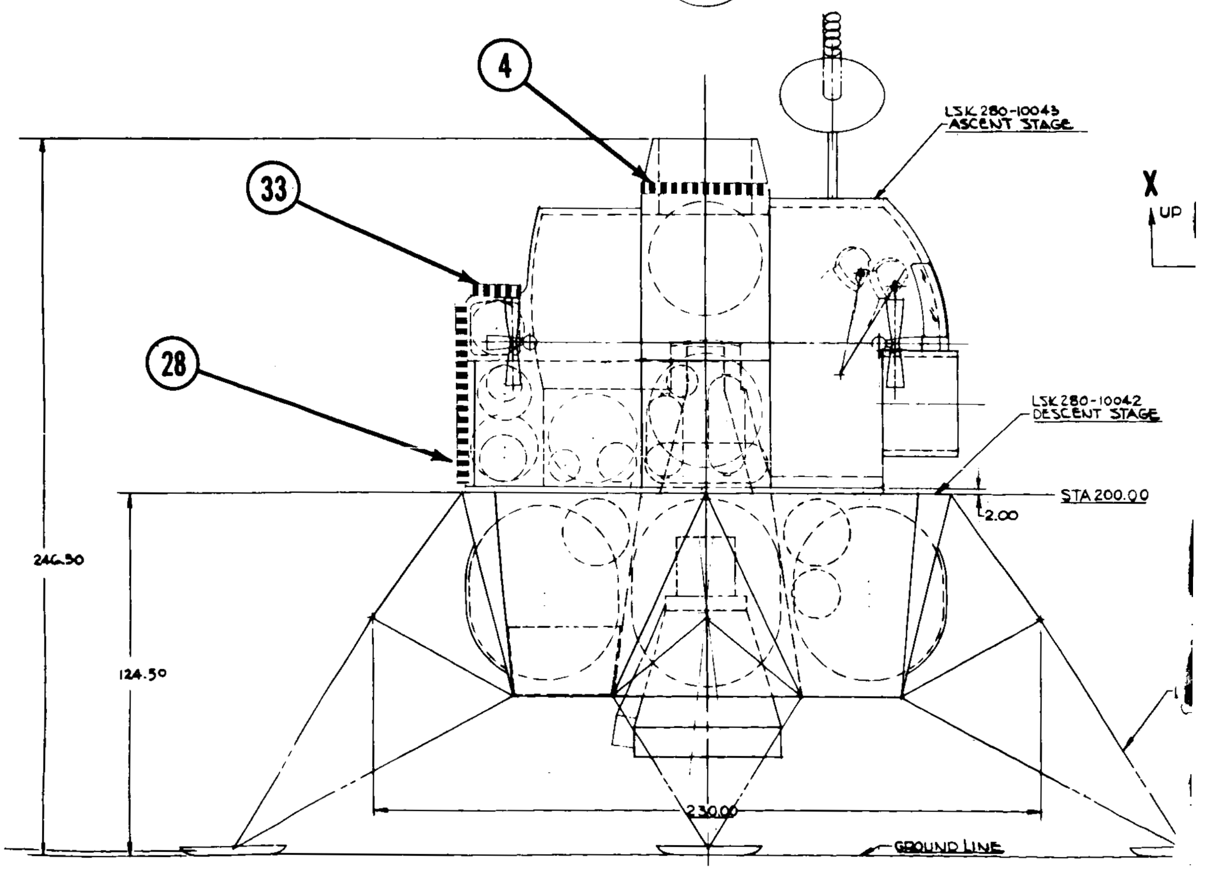

LOADING POSITION
SPACE SIMULATION CHAMBER
LN2 COLD WALL
HOISTING POSITION
LEM SPACE SIMULATION TEST

FIGURE 1





Contract No. NAS 9-1100
Primary Code No. 701

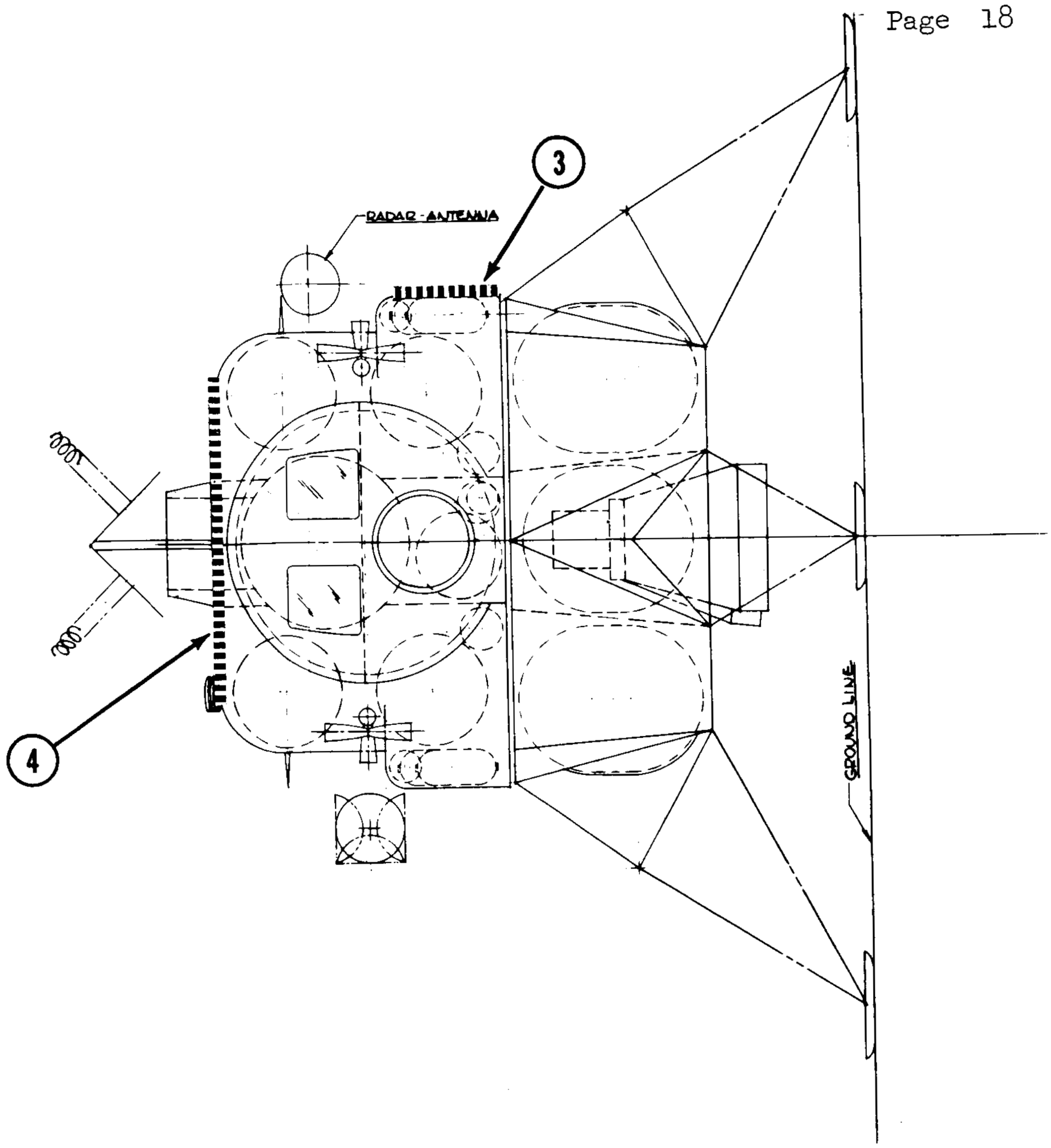


FIGURE 2

UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES. TOLERANCE UNLESS SPECIFIED FRACTIONS - DECIMALS - ANGLES	DRAWN BY	G. YEE 4-10-63	GRUMMAN AIRCRAFT ENGRG CORP BETHPAGE, L. I., N. Y.
	LAYOUT BY		
	CHECKED BY		GENERAL ARRANGEMENT - LEM
	GR LEADER		
ORIG DWG AUTH	STRUCTURE		
DETAIL SPEC REQUIREMENT	STRUCTURE		
CLASS II ENGRG CHANGE	WEIGHTS		CONTRACT NO
CLASS I ENGRG CHANGE	PROJ ENGR		CODE IDENT NO. 26512 · SIZE · LSK 280-10041
GRUMMAN REL INFO	GOVT APPD		SCALE 1/50 · SHEET

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